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Mars Ascent Propulsion System (MAPS)
Technology Program: Plans and Progress

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The return of rock and soil samples from the surface of Mars in the next decade is a focal point of the NASA Mars Exploration Program. In order to fit within the capability of medium-lift launch vehicles, it is necessary to utilize advanced propulsion technologies for the ascent from the Martian surface to Mars orbit. The ascent propulsion system has tremendous leverage on the injected mass requirements of the mission. The Jet Propulsion Laboratory (JPL) has embarked on a multi-year research and advanced development program to bring enabling propulsion technologies to prequalification status (TRL 6) by the end of GFY 2000. The propulsion technologies being pursued are: 1) ultralight propellant and pressurant tankage, 2) lightweight flow control devices, 3) low-temperature storable propellant combinations, 4) lightweight main engine and attitude control thruster design, and 5) warm gas pressurization systems. In addition, advanced development of packaging and structural concepts is ongoing. This paper will present an overview of the rationale for the technology selection, the program plans, and progress to date.

Introduction

The present technology task to develop Mars Ascent Propulsion System (MAPS) technologies is a result of mission studies conducted for the Mars Exploration and Technology and Applications Directorates at JPL. These studies¹ led to the selection of a set of high-payoff, relatively low-risk propulsion technologies to enable a Mars sample return mission to be performed in 2005 within the fiscal constraints of the Mars Exploration Program. Several technologies which have received a great deal of attention, such as in-situ propellant production, were rejected on the basis of modest payoff and/or excessive technical risk.

The Mars Exploration Program

In the wake of the loss of the Mars Observer Spacecraft², NASA and JPL conceived of a new approach to the exploration of the planet Mars based on frequent, lower-cost missions in place of the larger, more expensive, and necessarily less frequent Observer class missions. Objectives were to increase the robustness of the scientific enterprise to occasional failures as well as to create a dynamic, evolving program able to adapt to new discoveries and new challenges as they arise.

The Mars Exploration Program (MEP) which emerged, involves sending at least one spacecraft to Mars at every opportunity, or roughly every 26 months. Furthermore, the MEP is to be

accomplished under a level funding profile. The first of these missions is the Mars Global Surveyor, which is currently in orbit around Mars and which will complete aerobraking to its mapping orbit in the fall of 1998. The next set of missions³ *will be* the Mars Climate Orbiter and the Mars Polar Lander, scheduled for launch in December 1998 and January 1999, respectively. These missions are to be followed in 2001 with reflights of the Mars '98 engineering systems with the minimum modifications required to accommodate a different suite of scientific and engineering experiments, and to allow a different landing latitude for the lander. The launch opportunity in 2003 will feature a much larger lander capable of carrying an extensively instrumented rover which will make detailed *in situ* examinations of rock and regolith samples and demonstrate rock coring and sample collection technologies required to support the Mars Sample Return (MSR) mission. The '03 lander operations will be complemented by orbital *AP* observations by the European Space Agencies⁴, Mars Express Orbiter. The missions described above provide a logical progression of our knowledge of Mars and of our engineering tools to allow the launch of the first of a number of MSR missions, beginning in 2005.

The MSR Mission

As currently envisioned, the MSR mission in 2005 will involve the launch of an orbiter / Earth return vehicle on a Boeing Delta II (7925H) or equivalent

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launch vehicle and a launch of a lander package on a Delta III or equivalent launch vehicle. The orbiter will enter Mars orbit propulsively and perform aerobraking maneuvers to lower its orbit to the vicinity of 400 km by 300 km [Note: all specifics given herein are extremely preliminary and subject to change]. The lander package will directly enter the Martian atmosphere, with the exception of a small cruise stage which will be jettisoned shortly before entry. After entry, the lander system will deploy the heat shield and use parachutes to reduce decent velocity to the vicinity of 80 m/s. Approximately 2.5 km above the Martian surface, the parachutes will be released, and a final propulsive descent to a soft touchdown will be performed using a monopropellant hydrazine propulsion system on the lander. The system landed on the surface of Mars will consist of the lander, a science rover, and a Mars Ascent Vehicle (MAV). To accommodate the level funding requirement of the MEP, the lander design will be common to the design of the 2003 lander and the rover design will be common to the 2003 rover, meaning that the definition of interfaces to the MAV needs to be completed in time to support the 2003 system design.

Once landed, the rover will be deployed and will spend a period of approximately 3 months examining samples of Martian rocks and regolith for the purpose of collecting and returning samples of maximum scientific value. Heavy emphasis will be placed on searching for samples which have the maximum potential to reveal whether Mars may have had life at one point in its history. The scientific and philosophical implications of such a discovery would be enormous, and the evidence of past surface water on Mars, in conjunction with developments in the terrestrial biology of "extreme organisms" makes such a finding a distinct possibility. The problem is that many of the investigations required to make such a determination can not, as a practical matter, be miniaturized and automated such that they can be conducted on Mars; therefore, it is necessary to return samples to Earth where they can be examined in detail by hundreds of investigators using state-of-the art laboratory instrumentation.

The samples collected by the rover are transferred to the MAV, which is sitting on the lander. The MAV will then launch from the Martian surface to a near-circular orbit of approximately 300 km altitude. The MAV, as currently envisioned, is a two-stage vehicle comprised of the Mars Ascent Avionics (MAA), which also includes the sample and sample container,

and the Mars Ascent Propulsion System (MAPS), which is comprised of all of the propulsion hardware and primary and secondary structure of the MAV.

Once the MAV is in orbit, the orbiter will rendezvous and dock with it, transfer the sample container to an Earth entry, and perform a series of maneuvers to launch onto an Earthbound trajectory. Shortly before Earth entry, the orbiter will release the entry capsule, which will land in a location to be determined.

MAPS Requirements and Design

The design of the MAPS is driven by 7 major requirements (or more correctly, goals):

- Free-space equivalent $\Delta V \geq 4.3$ km/s
- Vehicle stack height ≤ 1 m
- Mars liftoff mass ≤ 600 kg
- MAA (payload) mass capability of 30 kg
- Minimal power and insulation mass requirements
- Aerodynamic fairing provided for ascent
- Maximum 3-year on-orbit lifetime

In addition to these requirements, the cost and risk of proposed technology developments were carefully considered.

MAPS Propulsion Technology Selection

The design trade studies which lead to the selection of the current baseline technologies have been previously documented¹ and will not be detailed again here. Instead, I will describe some of the key results qualitatively.

The use of In-Situ Propellant Production (ISPP) was found to provide essentially no benefit for this application. This was partially because the roughly 500-day period required to produce propellants on the surface constrained the arrival and return trajectories such that the available launch vehicle throw-weight was reduced. This is an aspect of the 2005 opportunity which would not necessarily be true for all launch opportunities, but even without this factor the study showed only a four percent reduction in injected mass (relative to the MAPS technologies under development) if the ISPP-compatible MAV used pressure-fed propulsion technology, and because of the trajectory constraints, this actually represents a reduction in injection mass margin. Another factor which caused the performance of the ISPP systems to suffer was that at least the oxygen produced on Mars had to be stored in a refrigerated liquid state, placing heavy power generation requirements on the lander.

In addition, cryogenic propellants in general suffered performance penalties due to their need for rather heavy propellant tank insulation and the low efficiency of pressure-fed feed systems caused by their low temperature. Cryogenic propulsion systems using LOX only became competitive when ISPP was combined with advanced lightweight pump-fed engine technology, and even then the differences in performance were within the probable error levels of the study. This level of technical risk was deemed unacceptable for such small payoffs.

Some exotic propellant combinations (such as fluorine / hydrazine) were very competitive on a mass basis but were not selected because of the high cost and technical risk normally associated with the use of halogenated propellants.

MAPS Design Description

The MAPS is a two-stage storable propulsion system which uses Monomethylhydrazine (MMH) fuel and Mixed Oxides of Nitrogen (MON) 25 oxidizer. The oxidizer is the reacted product of 25% (by weight) NO with 75% by weight N_2O_4 . This propellant combination was selected to provide a hypergolic, storable, system with a freezing point below -50°C , the approximate diurnal average temperature on Mars.

Figure 1 illustrates the configuration of the Mars Ascent Vehicle. The first stage uses two oxidizer tanks, two fuel tanks, and two pressurant tanks surrounding a biconic core structure. The first stage also features a conical fairing (not shown in Figure 1 for clarity). Two large (approximately 400 lbf) Main Engines Assemblies (MEAs) are used for the first stage firing, with four approximately 35 lbf Attitude Control Thrusters (ACTs) providing 3-axis attitude control.

The second stage uses four ACTs to provide both attitude control and primary propulsion. It also incorporates a stinger which, in the mated configuration, reaches to the base of the vehicle. This is where the sample container is mated to the second stage, probably by pyrotechnic welding. The second stage aerodynamic fairing is carried all the way to orbit, and is used as a substrate for solar cells which are needed to provide power while on orbit.

The feed system of the MAPS first stage is shown in Figure 2.

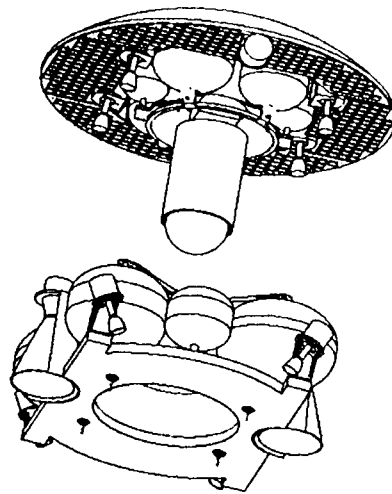


Figure 1 - Mars Ascent Vehicle Configuration

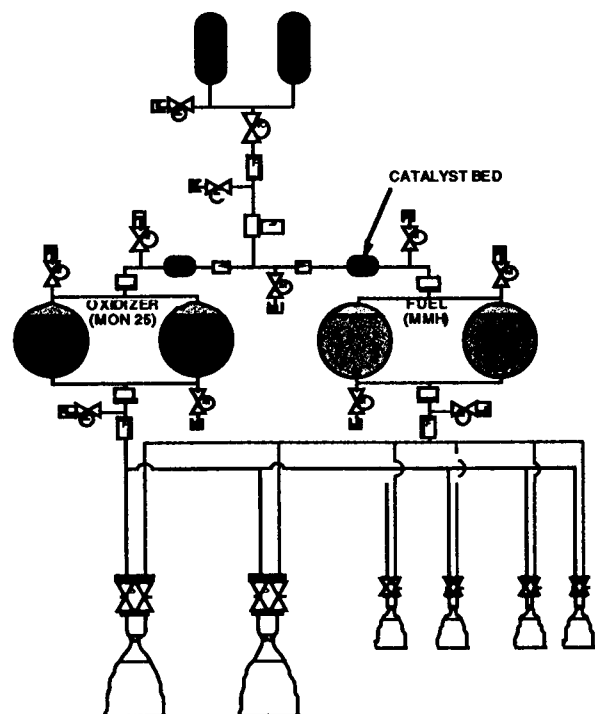


Figure 2 - MAPS First Stage Schematic

The system is activated shortly before liftoff by firing a single normally closed pyrotechnic valve. The pressurant gas then flows through a filter, the pressure regulator, and the fuel and oxidizer check valves and catalyst beds and begins to pressurize the system upstream of the pressurant side burst disks.

When the upstream pressure approaches the 500 psia regulated tank pressure, these pressurization burst disks rupture, allowing the propellant tanks to pressurize. When the tank pressures approach the regulated pressure, liquid-side burst disks rupture, priming the (previously evacuated) propellant lines. This approach provides for absolute propellant isolation until just before system activation without the mass and complexity of pyrotechnic valves or electronic latching valves.

The catalyst beds in the fuel and oxidizer sides of the MAPS pressurization system are designed to react a pressurant gas mixture of helium, hydrogen, and oxygen to produce warm helium and water vapor to pressurize the propellant tanks. This reduces the pressurant gas and tankage mass by an estimated 30% compared to a conventional cold-gas pressurization system.

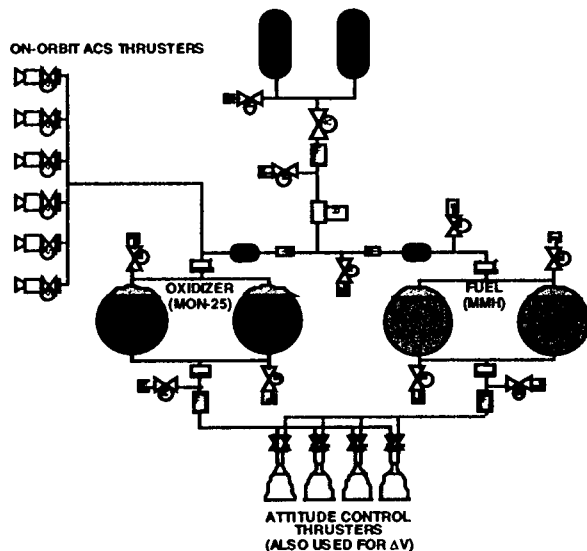


Figure 3 - MAPS Second Stage Schematic

Figure 3 shows the feed system schematic for the second stage of the MAPS. In addition to the features described for the first stage, the second stage includes a set of six 5 mN thrust cold-gas attitude control system (ACS) thrusters. These thrusters are fed from residual pressurant in the pressurant tanks and oxidizer tank ullages. Residual oxidizer vapor will also provide propellant to these thrusters. Once on orbit, these thrusters will orient the solar array toward the sun, spin the system to approximately 0.5 rpm, and maintain a crude sunpoint until shortly before rendezvous by the orbiter. At that time, the MAPS will revert to 3-axis control and become a passive participant in the terminal docking. This approach

limits the ACS thruster duty cycle to less than 1000 pulses total, greatly reducing the risk of excessive cold gas leakage depleting the available propellant prior to docking.

MAPS Technology Development

The technologies to be pursued for the MAPS system and early experimental results have been described elsewhere^{1,4}; this paper will discuss the approach to developing these technologies in more detail, including recent progress and near term plans.

Overall Approach

The MAPS technology effort began in FY'97 with funding of \$500K from the Mars Exploration Technology Program. During that year, the definitions of the technologies to be pursued were refined and preliminary proof-of-concept experiments were performed. It was shown⁴ that a conventional granular catalyst bed of Shell 405 could be made to function on a dilute mixture of helium, hydrogen, and oxygen at initial gas and bed temperatures below -50 °C. It was also shown that the MON-25/MMH propellant combination would experience hypergolic ignition at -40 °C, although the observed delay time of 30 to 60 ms is of some concern. Concepts for fabrication of ultralight composite overwrapped propellant and pressurant tanks were developed, and potential techniques for metalization of polymeric tank liners were demonstrated.

In FY'98 a three-year MAPS technology and advanced development program was launched when the \$500K/yr technology task was augmented by \$5M/yr from the Exploration Technology Program, a new line item in the NASA Code S budget created to "bridge the gap" between basic technology and flight system development. In recent decades, technology efforts for deep-space applications have often stopped far short of the point at which a flight system manager would be willing to accept the risk of adopting the new technology. This has had the unfortunate effect of greatly hampering the infusion of new technologies into flight projects. This is simply unacceptable as missions become increasingly demanding, and the Exploration Technology Program is one of a number of new initiatives to improve this situation. The basic MAPS approach is this:

1. The technology task performs basic studies such as material properties investigations, proof-of-concept experiments, and trade studies,

2. The advanced development task attempts to take the basic technology results and develop prototype hardware,
3. Prototypes of various new-technology hardware are integrated into a full-scale test bed and subjected to system testing of increasing fidelity, culminating in hot-fire system testing of a flightweight system at the end of FY '00,
4. Problems found during the development of prototype hardware and systems feed back into the basic technology program for resolution.

The objectives of this iteration of basic technology, hardware development, and integrated system testing is to allow high confidence in making flight technology selections by the end of FY'00. To meet this objective, it is necessary to have hardware tested and ready to go directly into a formal qualification program using flight system development funds. This requires substantially more development and testing than has traditionally been funded under technology programs; hence this task is indeed structured to "bridge the gap".

Warm Gas Pressurization

While it was shown during FY'97 that it was possible to obtain high reaction efficiencies using conventional granular packed catalyst beds, these beds tend to display significant and undesirable pressure drops, particularly at the high pressurant flow rates (over 30 scfm) expected for the MAPS first stage. They also tend to generate particulates during vibration and thermal cycling, which should be minimized in the MAPS design, especially on the second stage with its contamination sensitive cold gas thrusters. Thus, recent efforts have concentrated on monolithic bed designs such as that shown in Figure 4.

These monolithic beds are fabricated from strips of flat and corrugated metal which have had their surfaces coated with an alumina substrate onto which is "washed" a catalytically active material. The catalyst bed testing is conducted in the apparatus depicted in Figure 5.

The pre-chilled pressurant gas enters the test catalyst bed (in the upper left-hand side of Figure 5) from above. The catalyst bed is instrumented with thermocouples and three pneumatically-actuated valves and sample bottles allow samples of the reacted gas to be collected for chemical analysis.

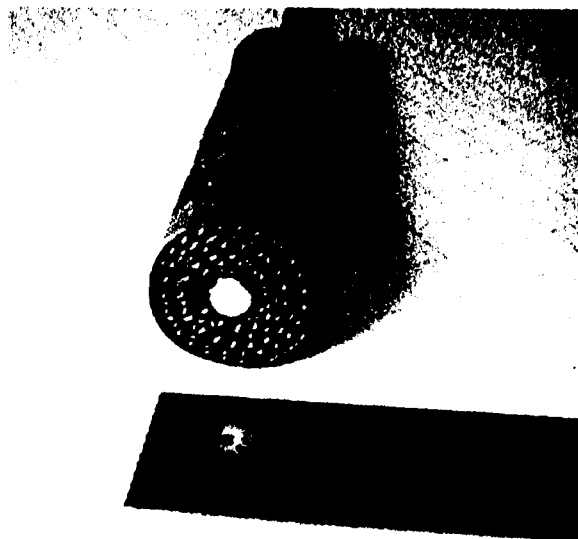


Figure 4 - Monolithic Catalyst Bed

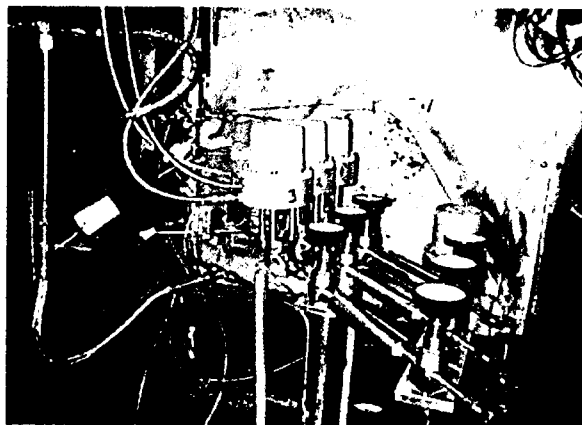


Figure 5 - Catalyst Bed Experimental Apparatus

The initial subscale testing was done with platinum catalyst material, and it was noted that the catalyst would not function at -40°C until it had been "activated" by operation from room-temperature and run for some time near 300°C . It is hypothesized that water, oxygen or some other agent deactivates the catalyst after long exposure to air. It was also found that the catalyst tended to be deactivated after repeated testing at low temperatures, requiring it to be re-conditioned by a room-temperature start. It is likely that this was caused by water vapor condensed within the system during runs contaminating the bed between runs; extra precautions to prevent this have been developed and are about to be tested. However,

low-temperature (-50 °C) runs with a freshly activated bed reliably produced temperature rises in close agreement with theory for the gas concentrations being used (primarily 3% hydrogen, 1.5% oxygen, and 95.5% helium). Unfortunately, the gas sampling and analysis methodology was still being worked out during this initial testing, so no quantitative measurements of reacted gas composition were made. The catalyst bed pressure drop was consistently below 1 psid at flow rates up to 7 scfm.

Testing conducted to date with iridium as the catalyst material have been discouraging. It was not possible to get a reaction started even at room temperature until the catalyst was activated by heating to over 200 °C in a reducing (hydrogen-rich) atmosphere. Reacted gas sample analysis indicated that the reaction between the hydrogen and oxygen was going only 80 to 85% of the way to completion. Even following activation, the bed showed no activity at -40 °C. However, the basic catalyst for these tests was residual material from testing done over a decade ago and was, quite literally, pulled out of someone's desk. A new sample, with better control on the iridium thickness and surface morphology, will be tested before this catalyst material is discarded for this application. However, at this point platinum appears to be the more attractive candidate.

Fabrication of housing parts for full-scale lightweight catalyst beds is in work to support a system pressurization and functional test to be conducted at JPL in September and to support full-scale propellant expulsion testing planned at the NASA White Sands Test Facility in August and September. The latter test is designed to test scale effects on heat transfer and interaction with propellants in the propellant tank ullage.

The interactions of the hot helium and water pressurant gas with the propellants takes two forms: decomposition of the MMH fuel and formation of corrosive nitric acid in the MON-25 oxidizer. The major concern with the MMH is the possibility of a runaway thermal decomposition, particularly if droplets of MMH are entrained into the ullage by the entering pressurant flow. The concern for the oxidizer is chemical attack on the propellant tank liners. This is not much of a concern for the first stage, since it is discarded in minutes, but is a significant concern for the second stage, which might have to survive three years in Mars orbit if the orbiter on the '05 mission fails. [In fact, on-orbit storage for up to three years is

baselined for follow-on sample return missions to reduce launch costs.]

Testing of the interaction of hot pressurant gas with MMH is just beginning. As precursors to this test, a study was made of the products of thermal decomposition of MMH at temperatures from 100 °C to 500 °C. These results suggest that there is little decomposition of ammonia, as would be expected at these low temperatures. These results will allow us to estimate the peak temperatures MMH sees in mixing regions during pressurant gas experiments. The hoped-for result is that we will find gradual and graceful degradation of the MMH in the ullage until it reaches completion, with no violent reactions. Subscale experiments will simulate inlet-to-surface spacing and surface-area-to-volume of the full scale testing.

Rocket Engines

The MAPS rocket engine requirements are unique in the use of MON-25/MMH propellants at -40 °C and the requirement to simultaneously minimize mass and maximize specific impulse. Some driving requirements for the ACT and MEA are given in Table I.

	<u>Mass</u> (kg)	<u>Isp</u> (s)	<u>Overall Length</u> (in)
ACT	0.75	310	9.5
MEA	5.5	325	22

Table 1 - Rocket Engine Requirements

In addition, the fact that Mars has a surface atmospheric pressure of about 10 millibar, or 0.15 psia, restricts the nozzle area ratio which can be used. Fortunately, a fairly high inlet pressure of 450 psia is provided to help make the engines lighter, smaller, and to allow use of higher area ratio nozzles.

During FY'97, JPL did ignition testing in a simple apparatus at -40 °C and characterized the viscosity of MON-25. The JANNAF rigorous performance methodology was used to set the Isp requirements shown in Table 1, assuming 95% combustion efficiency for the ACT and 98% for the MEA. During this summer, we expect to perform ignition testing of a simple injector in a quartz chamber with high speed video documentary of the results.

To begin advanced development of the ACT, JPL issued an RFP and made a competitive selection of two contractors for the initial phase of development of an injector. The contractors are GenCorp Aerojet and Kaiser Marquardt. Initial injector performance characterizations are to be performed by the end of September, with incorporation of lightweight combustion chamber technologies in the early fall if all goes well. The rationale for selection of two contractors for the initial phase was to reduce program technical approach by using two different technical approaches and to provide the maximum flexibility to respond to technical problems. The contracting approach for the advanced development of the MEA, expected to begin in FY'99, has not been defined at this time.

Since the MAPS performance requirements strive for both high performance and low mass, both rocket engine contractors have baselined a concept for a lightweight, high temperature combustion chamber fabricated from thin CVD layers of If/Re with a structural outer layer of Carbon-carbon proposed by Ultramet⁵. Both JPL and its contractors expect to continue to look at other options capable of meeting the MAPS requirements. NASA Lewis Research Center is planning to test lightweight combustion chambers fabricated by Ultramet and Refractory Composites, Inc.⁶ this summer. The data from this testing will be useful in evaluating these candidate chamber technologies for MAPS, although the chemistry of the combustion products will be different, limiting the direct applicability of the data.

Lightweight Components

Due to the relatively small size of the MAPS, the mass of components used to control fluid flow can be a major contributor to the dry mass of the entire system. This is especially true of the first stage because of the high thrust requirement, which requires pressurant and propellant flow many times greater than a typical spacecraft apogee boost system. Table 2 compares mass goals for MAPS components to the actual masses for the Cassini spacecraft, which had eight times lower flow requirements.

<u>Component</u>	<u>MAPS Mass</u>	<u>Cassini Mass</u>
	(g)	(g)
Pyro Valve	150	120
Service Valve	10	230
Gas Filter	150	400
Pressure Reg.	500	740
Check Valve	75	450
Liquid Filter	150	180
Burst Disk Assy.	150	520

Table 2 - Comparison of MAPS Component Masses to Cassini

JPL recently signed contracts with three contractors to develop lightweight component technologies for MAPS. Connax Florida Corporation is beginning development of the pyrotechnic valves, Moog Space Products Division is beginning development of the pressure regulator, and Vacco Industries is beginning work on the service valve, check valve, and a liquid filter. Preliminary design reviews had been held on the pyro valve, service valve, and check valve by the end of May. It is anticipated that work on lightweight components for the second stage will begin in FY'99.

In all cases, "proof-of-concept" units which display some or all of the functionality of the final lightweight component design are being delivered to JPL for incorporation into a full-scale testbed for system-level functional testing this September. These system-level tests are an annual event and will be of higher and higher fidelity, leading up to system hot-fire testing in FY '00.

Ultralight Tankage

Conventional propellant tanks for spacecraft are typically machined from titanium forgings, while conventional pressurant storage tanks are either titanium or a metallic liner (of titanium, stainless steel, or aluminum) overwrapped with a high-strength carbon composite. Tankage fabricated using these conventional technologies would be the largest single contributor to the dry mass of a MAPS. Twenty to thirty percent of the mass of high-pressure Composite Overwrapped Pressure Vessels (COPVs) used in conventional pressurant tanks is in the metallic liners which are required for acceptably low gas permeability. Attempts to apply COPV technology to low pressure propellant tanks have been hampered by the limitations of conventional liner fabrication technologies; a minimum thickness metallic liner can typically support a large fraction of the pressure load

and thus the benefit of the composite overwrap is greatly reduced. Also, the composite overwrap has a minimum ply thickness which is greater than that needed for structural performance in low pressure tanks which usually has resulted in excessive mass. The goal of this effort is to explore new liner fabrication technologies, fibers, and matrix materials which could produce factor of two reductions in propellant tank mass and reductions of 10 to 20 percent in pressurant tank mass.

Replacement of metal liners technologies which may allow thin-film metalization of the inside surfaces of fluorocarbon liners is one possible approach to reducing tank mass. It is hoped that if a truly amorphous metalized layer can be produced it would exhibit resistance to cyclic fatigue superior to that of conventional machined metal liners.

Limited permeability testing has been conducted on planar samples of Teflon PFA metalized by electroless plating and magnetron sputtering. Further testing will involve metalizing small Teflon sample bottles which can be overwrapped, then tested for permeability and cycle life.

In addition to investigation of polymeric liners, several potential technologies for forming very thin metallic liners on expendable mandrels are being pursued. These technologies could also prove viable for metalizing the outside surfaces of polymeric liners. Samples of a polyurethane foam mandrel material have been provided by Gail Gordon of the NASA Marshall Space Flight Center and appear to be a very viable mandrel material. Water-soluble clay mandrels are also being considered and Dr. Hoffman of the U.S. Air Force Research Laboratories has agreed to coat one of them with a candidate electroless plated liner material during the summer of '98. Extension of the technology used to form beverage cans from very thin metal foils is also being explored. The extension of existing techniques for chem-milling wrought aluminum liners to wall thickness below 0.005" is being carried as a lower-risk "fall back" technology..

Demonstrations of fabrication of tank liners will be followed by demonstration of prototype tank fabrication. In the propellant tank application, the use of polybenzoxazole (PBO) fibers in the overwrap will be investigated. In a minimum-lay-up application, the lower density of the PBO fibers may make them a superior material to the conventional T-1000 carbon fibers in spite of their slightly lower

strength. In addition, PBO is available in thinner plies than T-1000 graphite. Full-size prototype tanks will be fabricated for compatibility tests and use in MAPS testbed functional testing later this year.

Structures and Configuration

Detailed design of a the configuration concept is underway to support fabrication of the MAPS testbed this summer. Many options were considered in arriving at this core-structure based configuration. Among them were use of the aerodynamic fairing as a load-carrying shell structure; unfortunately, when we examined the local stiffening required to handle point loads from tank support struts, engines, etc., this concept was determined to be too heavy. The possibility of using propellant tanks as load-carrying members was considered especially attractive because the MAPS propellant tanks are launched from Earth with only a small pad pressure in the tanks, so that most of their strength is available to carry launch loads. However, carrying the loads through the tank bosses did not appear mass efficient because of the very high moment loads that could be applied to the structure near the bosses. Building tanks with integral composite skirts (which would also carry the hoop loads when the tanks were pressurized) looked very attractive, but did not seem to be mass efficient due to the fact that the minimum lay-up of the skirts significantly exceeded the required strength.

The central core structure selected for detailed design therefore appears to be the most mass efficient concept examined. The trade space of configuration options will be reexamined once detailed mass estimates for this design are completed. We will also factor in the results of a trade study on materials of construction and design concepts contracted to Composite Optics, Inc.

Conclusion

A combination of advanced storable propulsion technologies promises to enable a Mars sample return mission to be performed on a launch vehicle compatible with the funding profiles constraints of the Mars Exploration Program. These technologies are being pursued by the Jet Propulsion Laboratory in conjunction with numerous government and industrial partners

Acknowledgment

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